BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Application of Small Nuclear Rockets to Manned Planetary Missions - Case 720

DATE: April 16, 1968

FROM: H. S. London
A. L. Schreiber

ABSTRACT

The utility of small nuclear rockets for escape from or capture into high eccentricity parking orbits on manned planetary missions is investigated. Two representative missions were briefly analyzed: a 1977 triple-planet flyby and a 1982 Mars stopover/Venus swingby. In both cases it was assumed that nuclear propulsive stages and the spacecraft are first launched into a 48-hour period, 100 n.mi. periapsis Earth orbit by three-stage Saturn V-class launch vehicles, and nuclear rockets are then used for Earth escape. In the case of the stopover mission, high eccentricity orbits of between 12 and 48 hours period were also assumed at Mars, with both capture and escape propulsion provided by nuclear rockets.

It was found that optimum values of thrust/gross weight are as low as 0.05 and not higher than about .15 as a result of using high-eccentricity orbits, and that curves of payload fraction vs. thrust/gross weight are rather flat in the vicinity of the optimum. As a consequence, a common engine of between about 10,000 and 40,000 lbs thrust (depending on spacecraft weight) is appropriate for both missions.

The use of such engines could result in more payload for a given number of launches, as compared with high performance chemical cryogenic rockets--on the order of 10-15% for the flyby mission and 40% for the stopover mission. However, these figures are based on simplified and somewhat arbitrary assumptions regarding stage inert weights, etc. and should be regarded as tentative.

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MEMORANDUM FOR FILE

INTRODUCTION

Previously suggested applications for small nuclear rockets have included a launch vehicle upper stage for unmanned interplanetary missions, since such stages would be relatively small, and perihelion impulse stage for manned Mars landing missions, since gravity loss effects would be minimal for such an application. It is the purpose of this memorandum to demonstrate that such rockets can be used for main propulsion including Earth escape on manned planetary missions when highly elliptical parking orbits are used at the planets.

Two missions were investigated: a triple-planet flyby mission, and a 1982 Venus swingby/Mars stopover mission. In both missions it is assumed that the spacecraft and the nuclear propulsion stages are launched into a 48-hour period, 100 n.mi. perigee-altitude Earth orbit, implying the use of three-stage Saturn V-class launch vehicles.

A nuclear stage is used to inject onto the interplanetary trajectory following rendezvous, docking and checkout operations. In the case of the Mars stopover mission, nuclear stages are also used for the Mars capture and escape maneuvers; capture orbits of 100 n.mi. periapsis altitude and between 12 and 48 hours period were assumed. The problems of matching spacecraft modules and propulsion stages to particular launch vehicle capabilities were not included in this limited analysis.

Notation

a = Semi-major axis of parking orbit

g = Earth gravity

I_{sp} = Specific impulse

Notation (continued)

R_P = Perigee radius of parking orbit

T = Thrust

t = Engine operating time

V_{PE} = Perigee velocity of parking orbit

V_{PH} = Perigee velocity of the nominal hyperbola

V_∞ = Hyperbolic excess velocity at termination of burn

W_E = Engine weight

W_G = Gross weight at start of propulsive maneuver

W_i = Stage inert weight (excluding engines)

WIEO = Weight in Earth orbit

W_L = Payload weight

W_{MEM} = Mars excursion module weight

W_P = Propellant weight

W_{SC} = Spacecraft (mission module + Earth entry module) weight

ΔV = Velocity increment theoretically obtainable from expending the propellant in field free space

ΔV_I = Impulsive velocity increment required at perigee to produce a desired hyperbolic excess velocity

ΔV_L = $\Delta V - \Delta V_I$ = losses due to gravity

μ = Earth's gravitational constant

ANALYSIS

Method and Assumptions

The variation of payload/gross weight ratio (W_L/W_G) with thrust/gross weight ratio (T/W_G) was determined for each of the escape and capture maneuvers. Nuclear stage inert fraction (W_i/W_P) was assumed to be 0.2, and engine specific impulse 825 seconds. Two values of thrust/engine weight ratio (T/W_E), 3.0 and 4.0, were used.

The method employed was first to select a value of $\left(\frac{T}{W_G}\right)$ in the range of interest. The equations of motion were then integrated, using a gravity-turn as the essentially optimal steering function, until the terminal condition was reached. Burning time was minimized by iterative selection of the point on the parking orbit at which to start (always before perigee).

Minimal W_P is then computed from $W_P = t \frac{T}{I_{sp}}$. Then $W_E = \left(\frac{T}{\frac{T}{W_E}}\right)$, and $W_i = \left(\frac{W_i}{W_P}\right) W_P$. By definition, $W_G = W_L + W_P + W_i + W_E$, and the value of $\left(\frac{W_L}{W_G}\right)$ corresponding to the selected $\left(\frac{T}{W_G}\right)$ is computed.

It was assumed that the terminal condition of the finite burn involves only an energy level and is independent of the direction of the hyperbolic excess velocity vector.

Gravity Losses

In computing ΔV_I, the reference is a nominal hyperbola with a periapsis radius equal to that of the parking ellipse and a hyperbolic excess velocity magnitude corresponding to the terminal energy of the integration. The escape hyperbola is related to the nominal hyperbola only in energy level.

To determine ΔV_L we first solve the rocket equation, $\exp\,(\Delta V/Ig) = \frac{W_G}{W_G - W_P}$, for ΔV at the conclusion of the optimal burn. We then compute

$$\Delta V_L = V_{PH} - V_{PE}$$

$$= \sqrt{\frac{2\mu}{R_P} + V_\infty^2} - \sqrt{\frac{2\mu}{R_P} - \frac{\mu}{a}}$$

We then have $\Delta V_L = \Delta V - \Delta V_I$, which are the losses due to gravity.

The Earth escape maneuver for the triple-planet flyby mission was characterized by a V_∞ of .22 emos. This is specifically representative of the 1977 triple-planet flyby mission (Reference 1), but is also typical of other dual or triple-planet flybys and Mars single-planet twilight flybys when launch window effects are included.

The 1982 Mars capture/Venus swingby mission is characterized by V_∞'s of .15, .13, and .22 emos for Earth escape, Mars capture, and Mars escape, respectively, representing a mission duration of approximately 1-1/2 years (Reference 2). Unpublished analysis by H. S. London has shown that in this particular mission the arrival and departure asymptotes at Mars are oriented such that for impulsive maneuvers, ΔV penalties of less than 50 fps need be incurred for off-periapsis capture and escape impulses, over a wide range of Mars arrival and departure dates. Thus, this penalty was considered negligible in the present study. The effect of finite thrust levels on the ΔV penalties associated with directional escape from an elliptical orbit should be assessed, however, in any further study.

FLYBY MISSION

Payload ratio is shown as a function of T/W_G for the flyby mission in Figure 1. Corresponding gravity losses and engine operating time are given in Figure 2. The optimum thrust/gross weight ratio is about .10 (for either value of T/W_E) resulting in an injected payload of slightly more than two-thirds the gross weight in orbit, but near-maximum payloads are attained over a broad range of T/W_G.

As an example, consider a manned mission based on rendezvous of two three-stage Saturn V's in the specified 48-hour period orbit. The payload capacity of SA-516 to this orbit is roughly 110,000 pounds, so that the initial gross weight in orbit would be 220,000 pounds. The optimum thrust level of the nuclear Earth escape stage would, therefore, be about 22,000 pounds and a payload of about 150,000 pounds could be injected on a triple-planet flyby (or a single-planet Mars twilight flyby). Engine operating time at this thrust level would be about 34 minutes; doubling the thrust to 44,000 lbs would decrease operating time to 16 minutes while payload would be reduced 3% at most. If, on the other hand, engine operating time of one hour is allowable, a thrust level of only about 14,000 pounds would be adequate and the payload would still be within 2 or 3% of the optimum value. The optimum thrust levels scale as the initial weight in orbit, so that the desirable thrust levels would increase or decrease proportionately with the desired spacecraft weight.

By comparison, a cryogenic chemical injection stage with I_{sp} = 460 seconds and λ = 0.9 could inject a payload of about 59% of initial gross weight-in-orbit on the same mission, or about 13% less payload than the low-thrust nuclear stage.

MARS CAPTURE MISSION

Payload ratio versus thrust/weight ratio curves, and corresponding variations in operating time and gravity losses, are shown for the Mars capture mission in Figures 3 - 8.

The combination of a small ideal ΔV (i.e., impulsive, at perigee) required for the Earth escape maneuver ($\sim$3,600 fps) plus the highly eccentric Earth orbit results in a very low optimum value of T/W_G, about .060 - .075. Engine operating time is about 30 minutes at these thrust levels.

The Mars capture maneuver optimizes at T/W_G between about .07 and .10, depending on T/W_E and orbit period, with operating time no more than 40 minutes in this range. Optimum T/W_G for the Mars escape maneuver is higher, between about .10 and .15, because of the relatively large ΔV ($\sim$11,000 fps from a 48-hour period orbit) associated with the 1982 mission. Engine burn time is longer for this maneuver; about 55 minutes at T/W_G = .10 and about 35 minutes at .15.

More to the point is the fact that in all these maneuvers near-maximum payload ratio can be attained over a thrust/gross weight ratio range of at least .05-.15.

Mars Escape

The range of T/W_G of between .10 and .20 for Mars escape, which gives near-maximum payloads, is more than adequate with regard to engine life assuming that an operating time of an hour is permissible (see Figure 8). With T/W_G in this range, the payload ratio for Mars escape would be between .48 and .525 depending on the period of the capture orbit and, to a lesser degree, on the thrust/engine weight ratio.

The payload can be considered to consist of the spacecraft (mission module + Earth entry module) and a midcourse correction stage. Arbitrarly assuming a return leg midcourse ΔV budget of 1,000 fps (500 fps on both the Mars-Venus and Venus-Earth phases), midcourse stage mass fraction of .85 and I_{sp} = 390 sec, the midcourse stage would be about 9% of the spacecraft weight. The gross weight prior to Mars escape would therefore be about 2.1 - 2.2 times spacecraft weight.

The optimum escape thrust then would be in the range of about.2 - .3 times spacecraft weight. A reasonable range of spacecraft weights might be approximately 100,000 lbs on the high side with an 8 man crew, to as little as about 50,000 with a 3 man crew.

Thus, the best thrust range would be around 20,000 - 30,000 lbs with a heavy spacecraft, or 10,000 - 15,000 lbs with a light spacecraft.

Mars Capture

Optimum T/W_G for Mars capture is lower than for escape, around .05 - .10, due to the lower V_∞ for Mars capture, and again the payload - T/W_G curve is very flat (Figure 5). Engine operating time is less than one hour even for T/W_G as low as .05 (Figure 6).

The payload/gross weight ratio for the capture maneuver is between about .73 and .765. "Payload" for the capture maneuver now consists of the Mars escape stage, the spacecraft, and a Mars excursion module (or unmanned probes). It was previously determined that the Mars escape stage would be 2.1 - 2.2 times the spacecraft weight; therefore the gross weight prior to the start of the capture maneuver, including the Mars capture stage weight, can be expressed as

$$W_{\text{Mars capture}} = a\,[bW_{sc} + W_{MEM}] = a\,W_{MEM} + a'\,W_{sc}$$

where a = 1.31 - 1.37 and b = 2.1 - 2.2 so that

$$W_{\text{Mars capture}} = (2.75 - 3.01)W_{sc} + (1.31 - 1.37)W_{MEM}\ .$$

Again taking as a "heavy" spacecraft one which weighs 100,000 lbs and a corresponding MEM which also weighs = 100,000 lbs, gross Mars capture weight would be between 406,000 and 438,000 lbs. Since the optimum value of T/W_G was about .07 - .10, the optimum thrust level would therefore be 28,000 - 44,000 lbs. For a small light S/C (W_{sc} and W_{MEM} 50,000 lbs each) the best thrust level would be about 14,000 - 22,000 lbs.

Earth Escape

The payload/gross weight ratio is about .81 for the Earth escape maneuver for values of T/W_G between about .05 - .10. The Earth escape "Payload" consists of the Mars capture and escape stages as well as the spacecraft, Mars excursion module, and midcourse propulsion --i.e., it is equal to the gross weight just prior to the Mars capture maneuver plus the Earth-Mars midcourse propulsion.

A midcourse ΔV of 500 fps is assumed for the Earth-Mars phase of the mission; the midcourse propulsion system will then weigh about .046 times the Mars capture gross weight. The gross weight in the elliptical Earth orbit (not including any items jettisoned prior to initiating the escape maneuver) is then

$$\text{WIEO} = \frac{1.046}{.81} \times W_{\text{Mars capture}} \approx 1.3 \times W_{\text{Mars capture}}$$

The gross weight prior to Mars capture was previously estimated at 406,000 - 438,000 lbs for a "heavy" spacecraft and MEM (100,000 lbs each) or half that for "Light" spacecraft and MEM (50,000 lbs each). Gross weight in Earth orbit is then about 530,000 - 570,000 lbs for the "heavy" weights, and half that for the "light" weights. The best thrust level for Earth escape is therefore anywhere from about 26,000 lbs to 56,000 lbs with the "heavy" spacecraft or 13,000 - 28,000 lbs with "light" spacecraft.

Mars Capture Mission - Summary

The "best" thrust levels (i.e., near-optimum in terms of gross weight, with engine operating times $\leq$ one hour) are summarized in the following table, where "heavy" and "light" spacecraft weights are arbitrarily assumed to be:

"Heavy"- mission module + Earth entry module = 100,000 lbs;
- Mars excursion module = 100,000 lbs;

"Light"- 1/2 the heavy values.

Mission Phase	Thrust Levels (lbs) "heavy" S/C	"light" S/C
Earth escape	26,000-56,000	13,000-28,000
Mars capture	20,000-44,000	10,000-22,000
Mars escape	20,000-40,000	10,000-20,000

If one wishes to select an engine which is appropriate for use with either "light" or "heavy" spacecraft, something in the 20,000 lb thrust category would seem suitable. Single engine configurations would then be appropriate for all three stages with a "light" spacecraft, as well as for the Mars capture and escape stages with a "heavy" spacecraft. In the "heavy" spacecraft case, the Earth escape stage would use two engines; the Mars capture and escape stages could also be two-engine rather than single engine configurations. Engines as small as 10,000-15,000 lbs thrust would be suitable for 2-4 engine stages with a "heavy" spacecraft and one or two-engine stages with a "light" spacecraft.

If an engine is selected on the basis of a "heavy" spacecraft only, then the preferred thrust level would be higher, say 30,000-40,000 lbs. This would allow single-engine configurations in all three stages and would take advantage of a presumably higher thrust/engine weight ratio as compared with a smaller engine.

Comparison with Chemical Stages

The initial weight required in the same 48-hour period Earth orbit if cryogenic chemical rather than nuclear stages are used was calculated for comparison. An I_{sp} of 460 seconds and λ of 0.9 was assumed for all stages. The same assumptions were made for midcourse ΔV and propulsion system characteristics as in the nuclear case.

It was assumed that the thrust/weight ratios for the chemical stages are sufficiently high so that gravity losses can be neglected. The results are:

$$WIEO = 2.4\ W_{MEM} + 5.89\ W_{s/c}$$ for a 12-hr. period Mars orbit

$$= 2.02\ W_{MEM} + 5.33\ W_{s/c}$$ for a 48-hr. period Mars orbit

For the "heavy" 100,000 MEM and S/C, therefore, the WIEO is between about 735,000 and 803,000 lbs. This compares with the estimate of 530,000-570,000 lbs using nuclear stages, indicating almost a 30% reduction in WIEO through use of small nuclear rockets, based on the assumed performance parameters. Conversely, these figures imply that using nuclear rockets will result in approximately 40% greater payload (i.e., total spacecraft weight) for a given number (and weight) of launches.

CONCLUSIONS AND REMARKS

1) The ratio of nuclear rocket thrust to vehicle gross weight which maximizes the payload/gross weight ratio for missions based on highly eccentric parking orbits at Earth and Mars is as low as .05 and not higher than about .15. The optimum value of T/W_G for a particular escape or capture maneuver increases as V_∞ increases, as the period (or eccentricity) of the parking orbit decreases, and as the ratio of thrust to propulsion system weight increases.

2) The masses which must be accelerated (or decelerated) during escape (or capture) maneuvers are much smaller when elliptical rather than low altitude circular parking orbits are employed This fact combined with the low values of optimum thrust/gross weight ratio associated with highly elliptical parking orbits means that much lower thrust magnitudes are appropriate than is the case when low-altitude circular orbits are used.

3) The curves of payload ratio vs. thrust/gross weight ratio are rather flat in the vicinity of the optimum, so that in practice there is a wide range of thrust level which will give near-maximum payload ratio for any given maneuver. This facilitates the choice of a common engine for different missions and mission phases.

4) The missions analyzed and the assumptions made in this brief study lead to the conclusion that a thrust level of between about 10,000-40,000 lbs per engine would be most appropriate for nuclear rockets which are used to inject a manned planetary vehicle from a highly elliptical Earth orbit into a transplanetary trajectory, and for Mars capture and escape stages as well.

5) A comparison with high performance chemical cryogenic stages indicates that the use of small nuclear rockets might significantly reduce the weight required initially in the elliptical Earth parking orbit for a given spacecraft weight (or the converse) --on the order of 10-15% for flyby missions and 25-30% for a Mars capture/Venus swingby mission. This conclusion is tentative, however, since it is based on somewhat arbitrary assumptions concerning stage inert weights, does not include separate consideration of boil-off, insulation, and meteoroid shield penalties, aerodynamic shroud weights required during launch, constraints on launch vehicle payload length, etc.

6) The results of this study do not necessarily mean that a small nuclear engine in the category considered is better than a larger engine in the Nerva I class ($\sim$75,000 lbs thrust) for manned planetary missions. On the contrary, higher thrust levels facilitate using nuclear stages to escape from a low altitude circular rather than highly elliptical Earth orbit, and would therefore result in better overall performance (i.e., more payload for a given number of launches) because more of the total ΔV is imparted by the high I_{sp} nuclear stages and less by the chemical launch vehicles. (Performance studies by NASA contractors have shown that still better performance would result if the nuclear stages are started suborbitally.) Similarly, higher thrust levels are better for attaining low altitude orbits at Mars or Venus. A second but less important advantage of high thrust engines is somewhat higher thrust/engine weight ratio.

On the other hand, numerous studies have shown that there is a large reduction in the WIEO for planetary capture missions if highly eccentric orbits are employed at the target planet, although some compromise of mission objectives may be involved, and the results reported herein clearly indicate that a small engine is better in that case for Mars capture and escape. The results furthermore show that such an engine could usefully be employed for Earth escape by starting from a high eccentricity Earth orbit into which the spacecraft and propulsive stages would be launched by three-stage Saturn V-class launch vehicles. The question of what would be a "best compromise" thrust level for missions using nuclear rockets to escape from low altitude Earth orbit and involving a high-eccentricity capture orbit at Mars or Venus has not yet been addressed. Neither has the feasibility of clustering large numbers (e.g., 6 or more) of small engines or of using multi-engine parallel stages been investigated for the Earth escape phase.

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1013-$\frac{\text{HSL}}{\text{ALS}}$-sjh

Attachments

REFERENCES

1. VanderVeen, A. A., "Families of 1977 Triple-Planet Flybys", Bellcomm Memorandum for File, June 6, 1967.

2. VanderVeen, A. A., "Venus Swingbys for Manned Mars Missions During the 1978 - 1986 Period", AAS Science and Technology Series, Volume 11, pp. 433-450.

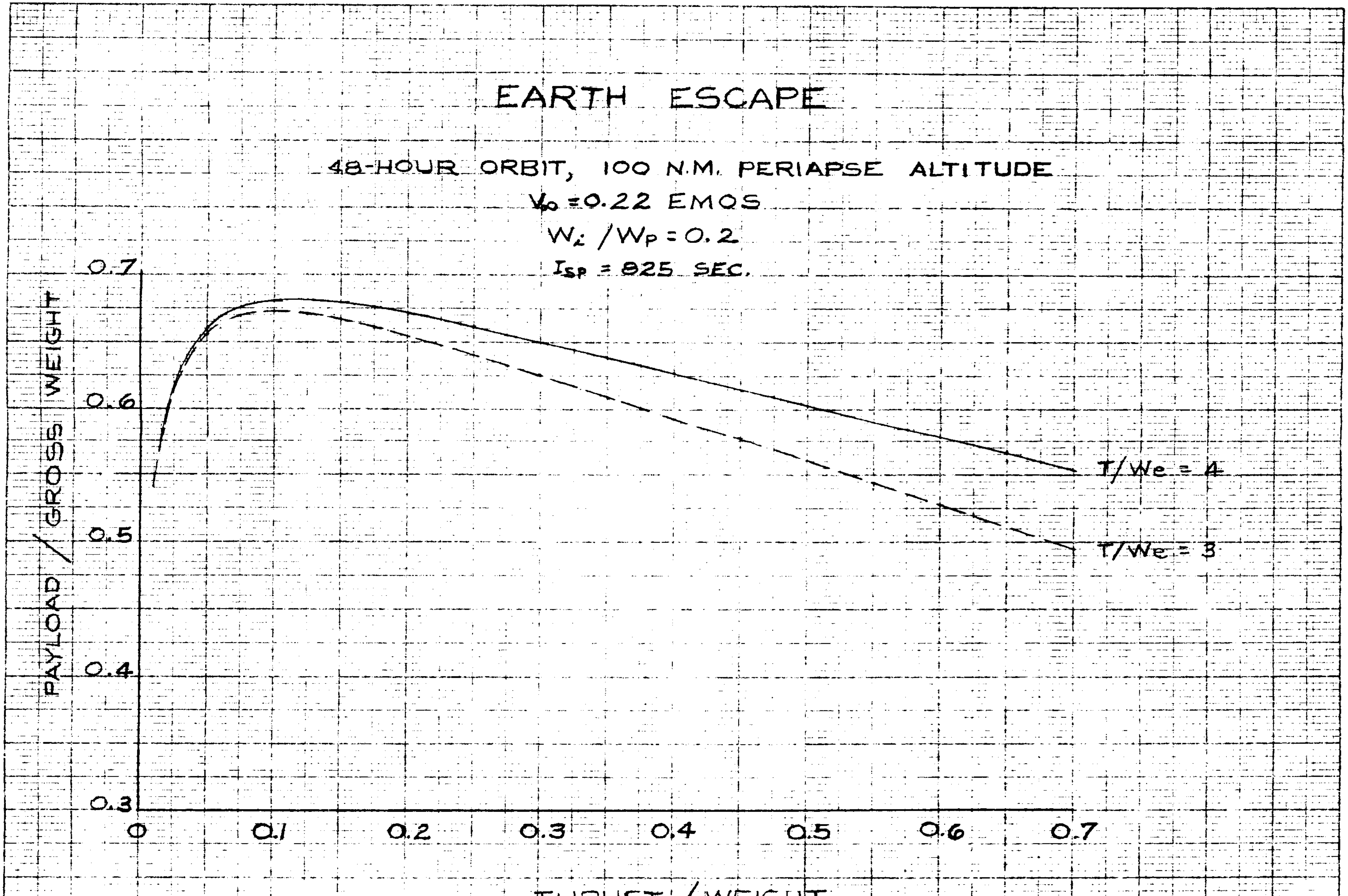
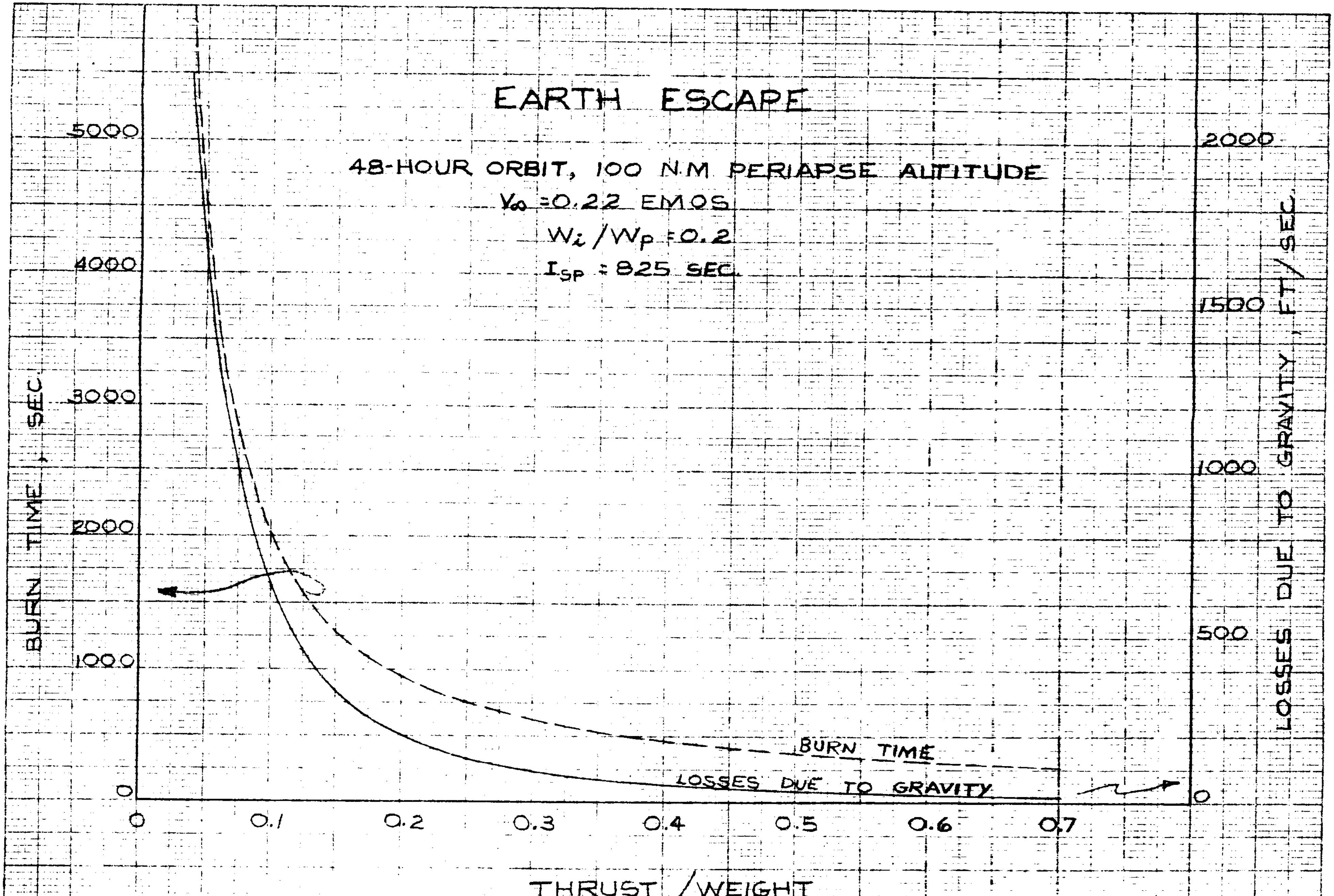

EARTH ESCAPE
48-HOUR ORBIT, 100 N.M. PERIAPSE ALTITUDE
V_0 = 0.22 EMOS
W_i / W_P = 0.2
I_{SP} = 825 SEC.
0.7
0.6
0.5
0.4
0.3
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
T/We = 4
T/We = 3
PAYLOAD / GROSS WEIGHT
THRUST / WEIGHT

FIGURE 1



EARTH ESCAPE
48-HOUR ORBIT, 100 N.M PERIAPSE ALTITUDE
V_∞ = 0.22 EMOS
W_i / W_p = 0.2
I_{SP} = 825 SEC
BURN TIME, SEC
5000
4000
3000
2000
1000
0
LOSSES DUE TO GRAVITY, FT/SEC
2000
1500
1000
500
0
BURN TIME
LOSSES DUE TO GRAVITY
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
THRUST / WEIGHT

FIGURE 2

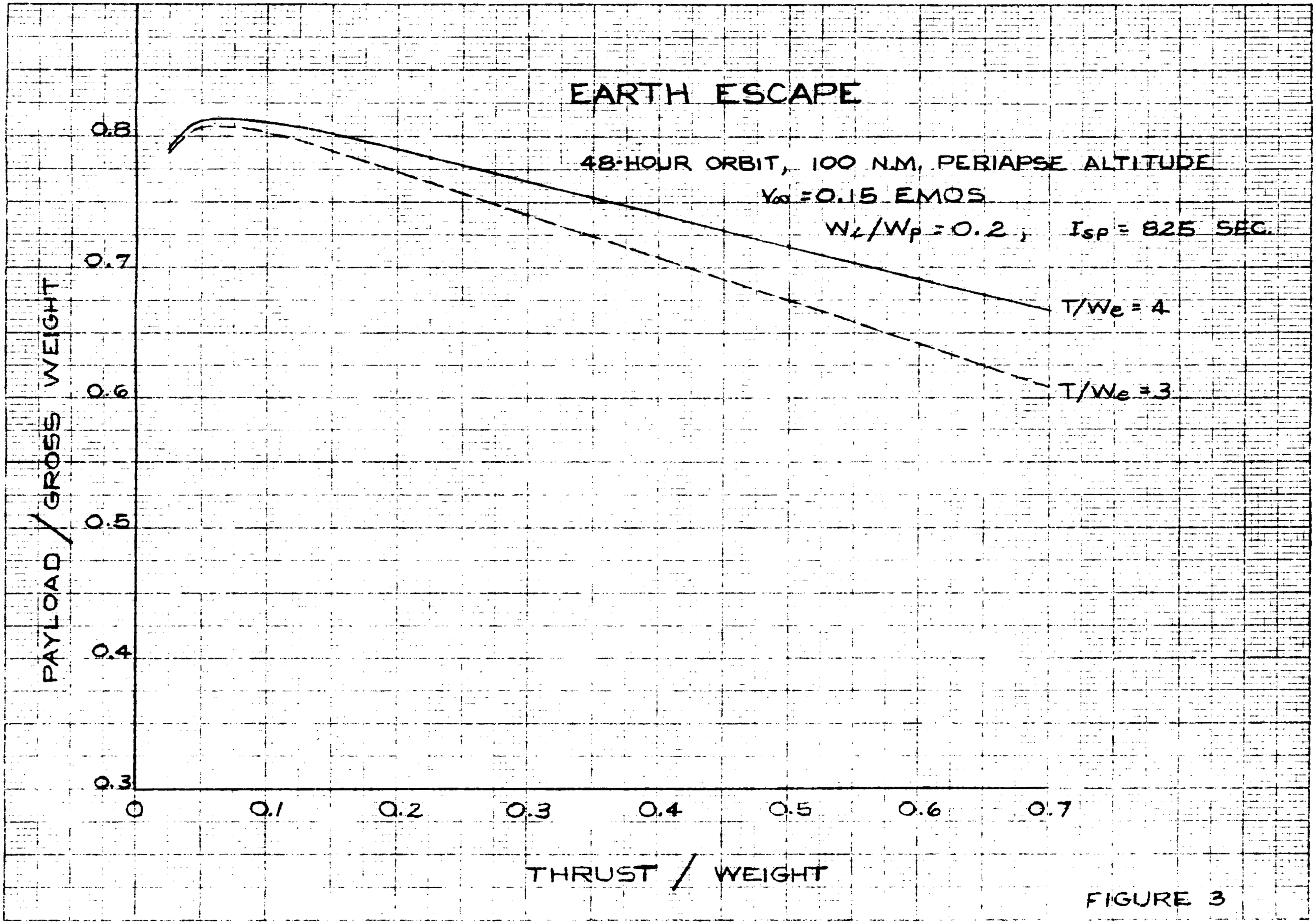
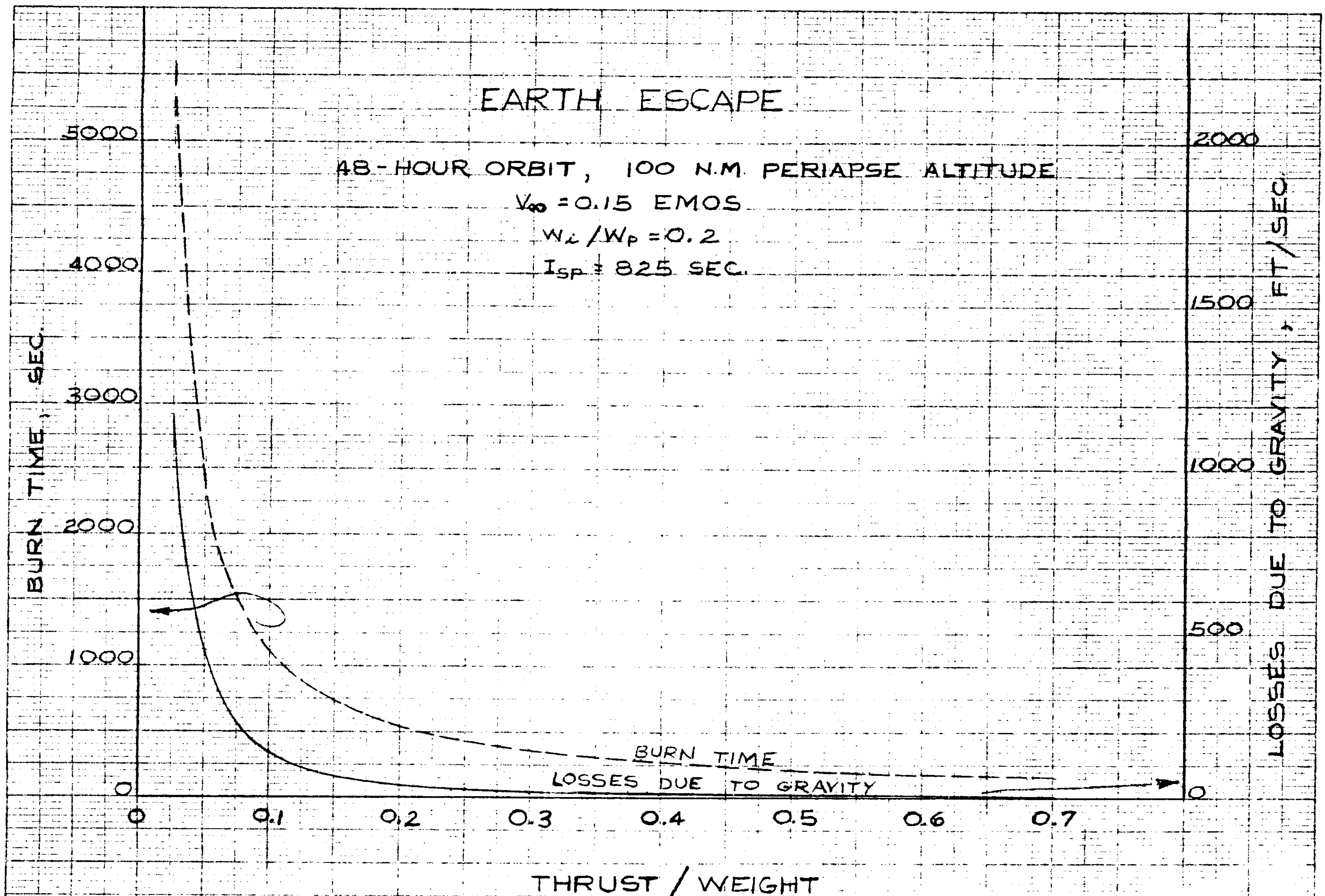
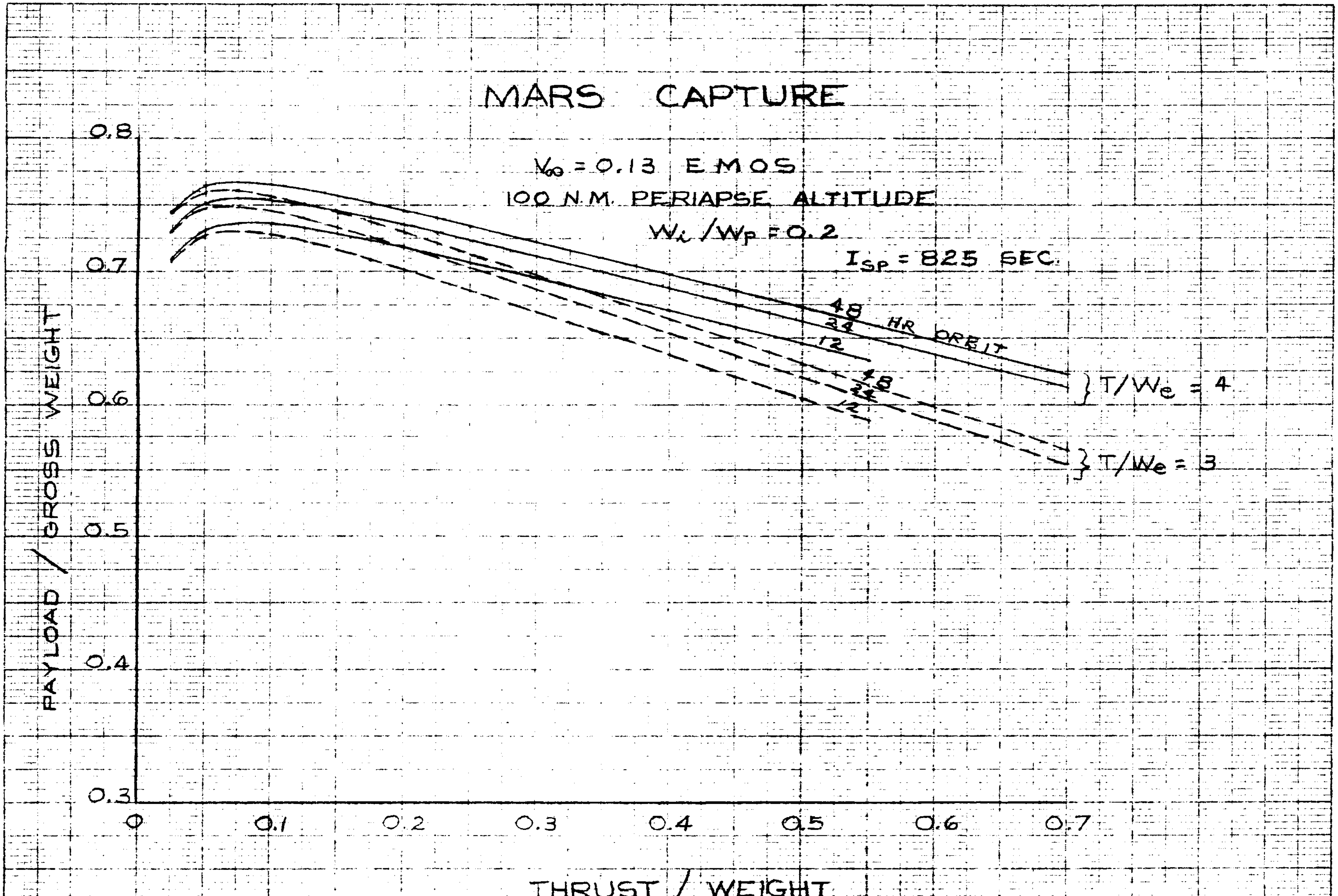

EARTH ESCAPE
48-HOUR ORBIT, 100 N.M. PERIAPSE ALTITUDE
V_∞ = 0.15 EMOS
W_L/W_P = 0.2 , I_{SP} = 825 SEC.
T/We = 4
T/We = 3
PAYLOAD / GROSS WEIGHT
0.8
0.7
0.6
0.5
0.4
0.3
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
THRUST / WEIGHT

FIGURE 3



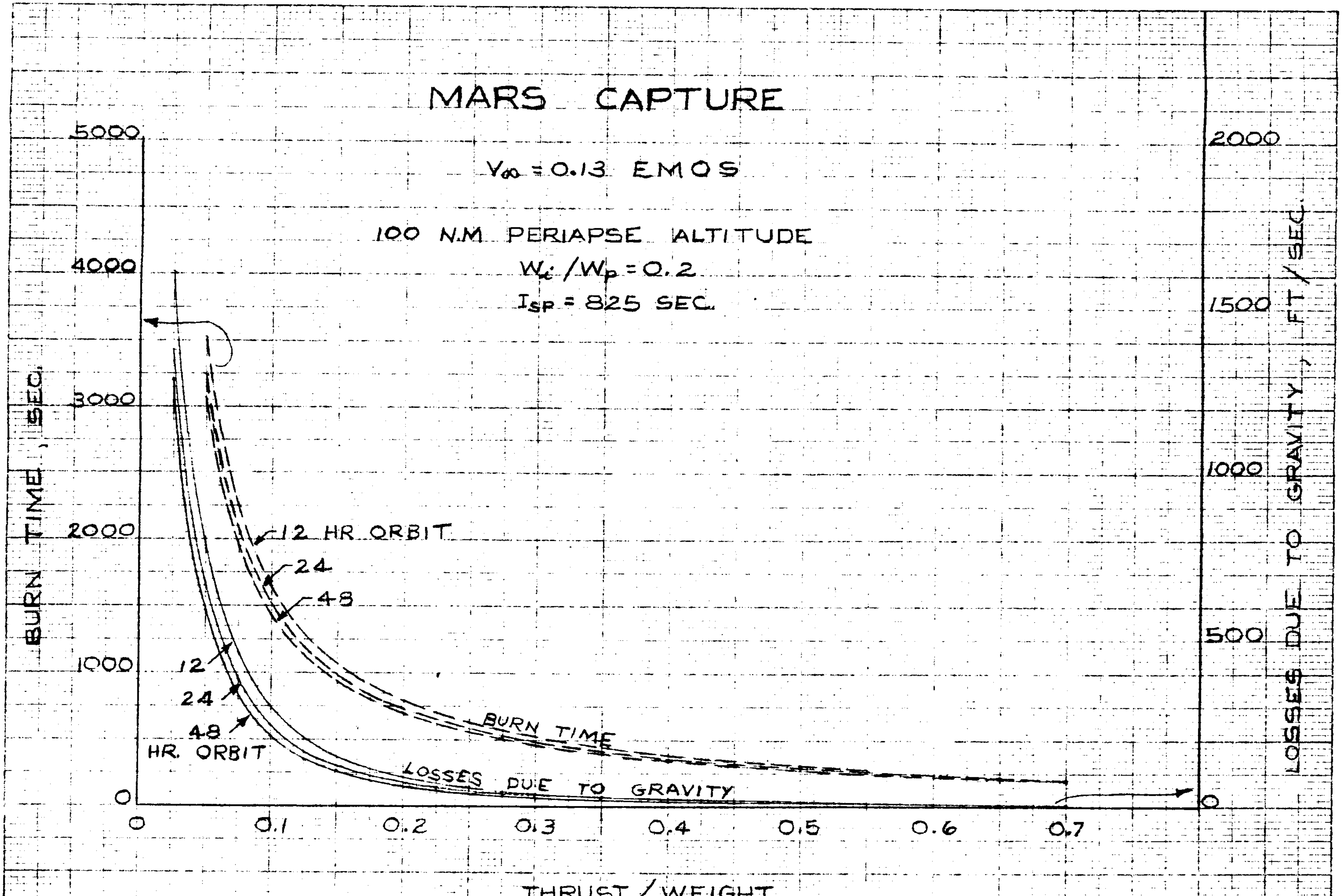
EARTH ESCAPE
48-HOUR ORBIT, 100 N.M. PERIAPSE ALTITUDE
V_∞ = 0.15 EMOS
W_i / W_p = 0.2
I_{sp} = 825 SEC.
BURN TIME, SEC.
5000
4000
3000
2000
1000
0
LOSSES DUE TO GRAVITY, FT/SEC
2000
1500
1000
500
0
BURN TIME
LOSSES DUE TO GRAVITY
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
THRUST / WEIGHT

FIGURE 4



MARS CAPTURE
V_∞ = 0.13 EMOS
100 N.M. PERIAPSE ALTITUDE
W_i / W_p = 0.2
I_{SP} = 825 SEC.
48 HR ORBIT
24
12
48
24
12
T/We = 4
T/We = 3
0.8
0.7
0.6
0.5
0.4
0.3
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
PAYLOAD / GROSS WEIGHT
THRUST / WEIGHT

FIGURE 5



MARS CAPTURE
V_∞ = 0.13 EMOS
100 N.M. PERIAPSE ALTITUDE
W_i / W_p = 0.2
I_{SP} = 825 SEC.
BURN TIME, SEC.
5000
4000
3000
2000
1000
0
LOSSES DUE TO GRAVITY, FT / SEC
2000
1500
1000
500
0
12 HR ORBIT
24
48
12
24
48
HR. ORBIT
BURN TIME
LOSSES DUE TO GRAVITY
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
THRUST / WEIGHT

FIGURE 6

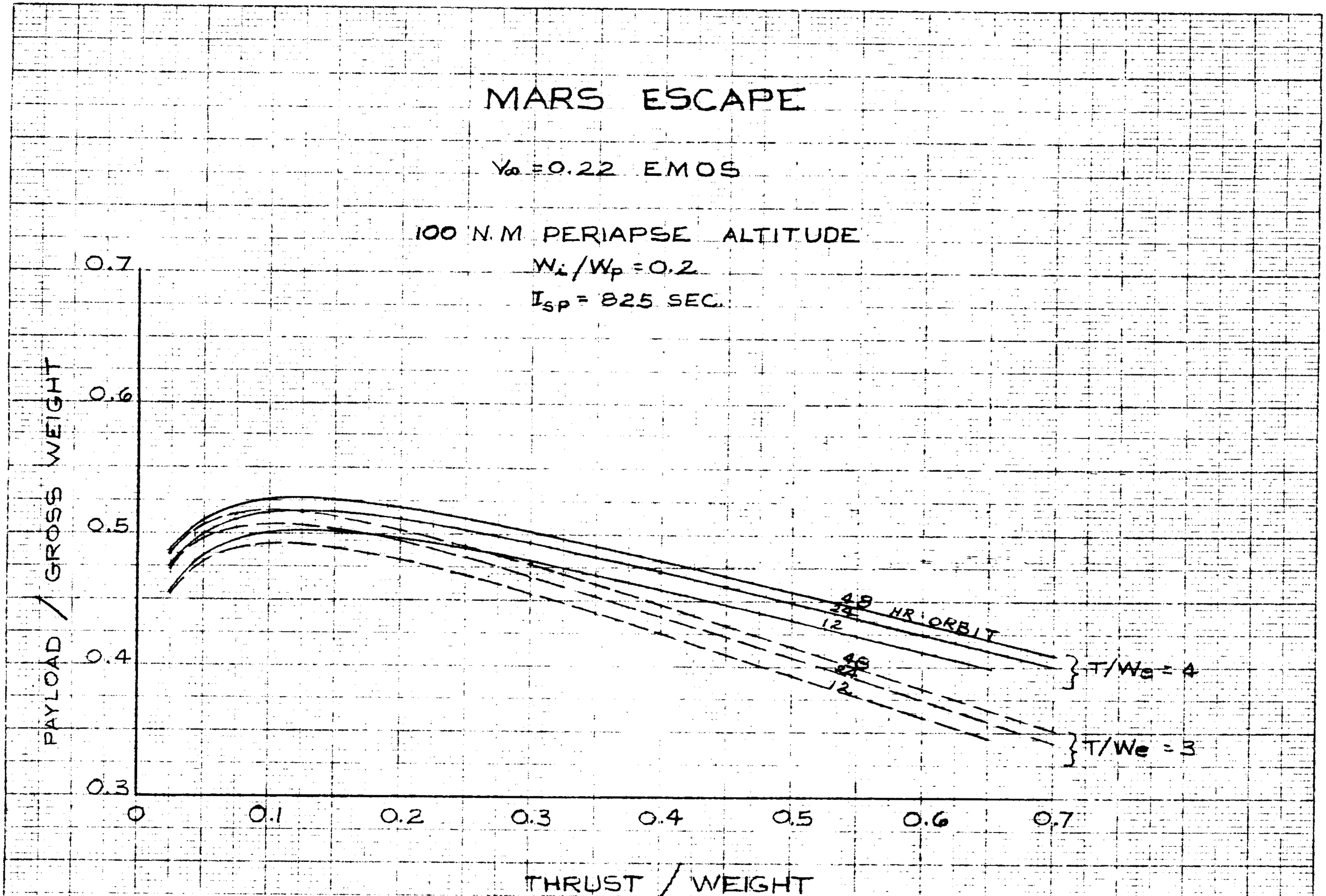

MARS ESCAPE
$V_\infty = 0.22$ EMOS
100 N.M PERIAPSE ALTITUDE
$W_i / W_p = 0.2$
$I_{sp} = 825$ SEC.
PAYLOAD / GROSS WEIGHT
0.7
0.6
0.5
0.4
0.3
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
48
24
12
HR ORBIT
$T/W_e = 4$
$T/W_e = 3$
THRUST / WEIGHT

FIGURE 7

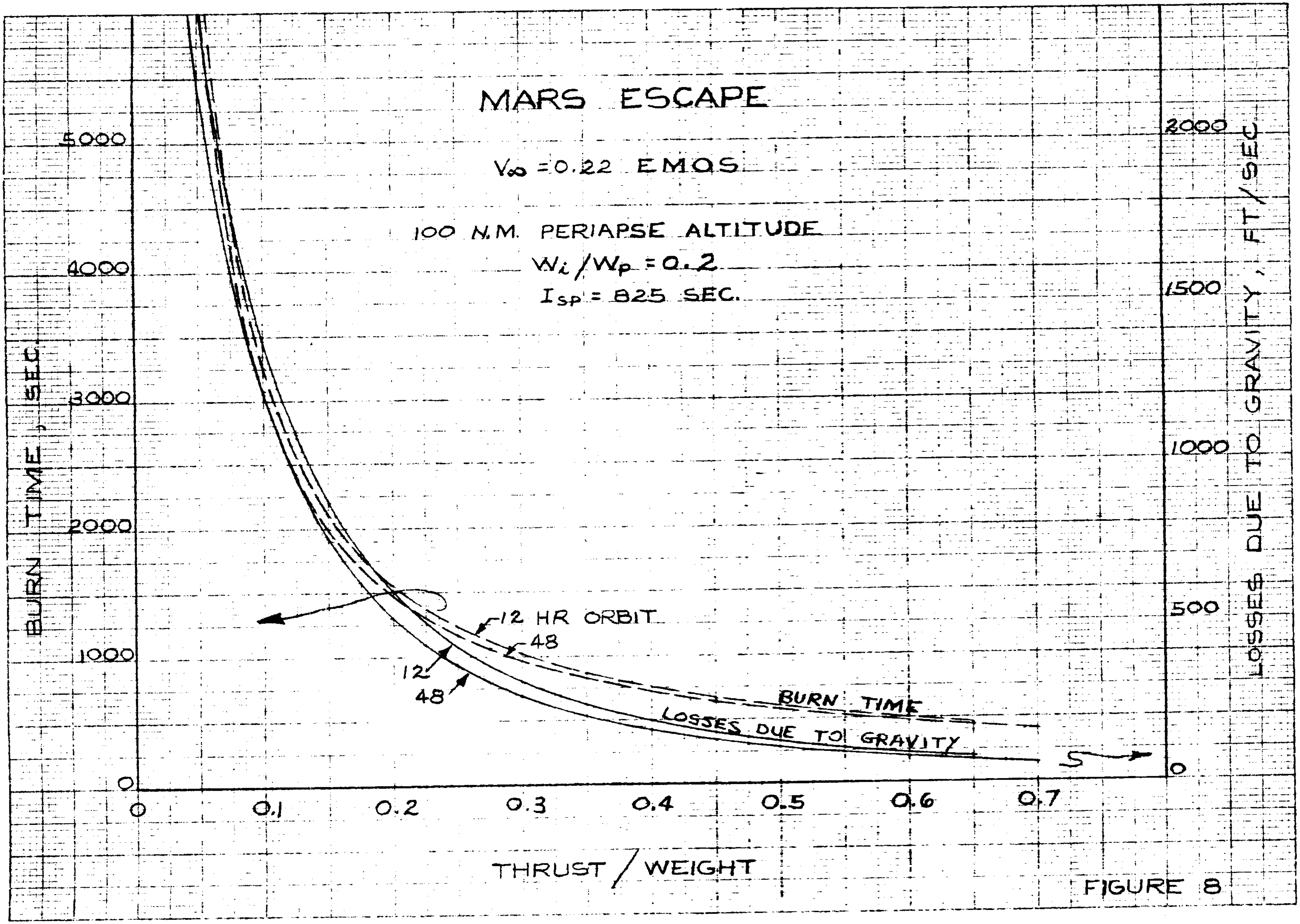

MARS ESCAPE
V_∞ = 0.22 EMOS
100 N.M. PERIAPSE ALTITUDE
W_i/W_p = 0.2
I_{sp} = 825 SEC.
12 HR ORBIT
48
12
48
BURN TIME
LOSSES DUE TO GRAVITY
BURN TIME, SEC
5000
4000
3000
2000
1000
0
LOSSES DUE TO GRAVITY, FT/SEC
2000
1500
1000
500
0
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
THRUST / WEIGHT

FIGURE 8

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From: H. S. London
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